V. **Space Weather Applications**

Although the 0.48 AU orbit chosen here may not be as ideal for space weather observations as a 90° inclination orbit at 1 AU, which would always be within 30° of the plane normal to the Earth-Sun Line, the 3 to 1 resonance orbit discussed in this report does put SPSM in a position to observe CMEs directed towards Earth the majority of the time. The instrumentation on SPSM could therefore make an important contribution to space weather forecasts. In particular, SPSM would be able, much of the time, to view the development of active regions on the back side of the Sun from Earth, thereby providing the potential for longer term forecasts. From its position at 0.48 AU, SPSM would, at times, also be in a position to observe large, gradual solar energetic particle events before the nose of the shock acceleration region crosses field lines connected to Earth, and it might therefore provide up to a day's warning of large solar proton events.

For the science observations alone, spacecraft operations can be straightforward and routine, simply repeating measurements at a cadence that is either predetermined by ground command or computed on board in real time. For science, a single telemetry session per week would suffice. There are two alternatives for the more frequent communications required for space weather warnings of a CME or energetic particles headed toward Earth. The first is to have continuous low-rate telemetry to a set of dedicated, nearly autonomous ground stations. As indicated in the previous Section, a rate of ~10 bps is adequate for detecting the occurrence of Earthward-moving CMEs and/or of increased energetic proton fluxes.

The second option, which has been analyzed in a bit more detail and costed, is based on currently evolving beacon-mode technology. In such a scenario, a simple two-level tone is used to indicate whether or not an event has occurred, and upon detection of the occurrence of an event, a large Deep Space Net antenna is used to acquire data on the its nature. Use of a beacon mode is cost effective if all of the following conditions are met:

- The phenomena being observed are dynamic and cannot be predicted.
- The occurrence frequency of the phenomena is low.
- The period of observation is long (e.g., years)
- The required response time or data latency is short (e.g., hours).

Institution of a beacon mode requires an onbaord telecom system that can communicate with Earth 24 hours/day and at least 3 ground antennas (LEO-T, 5-meter stations, one at each DSN site, estimated cost $2M) integrated into the DSN capabilities (estimated cost $1M per station). Upon detection of an event, emergency use of a 70-m DSN antenna would command the spacecraft to transmit a special downlink load which might consist of 4 compressed images of the event together with a modest amount of energetic particle data. The $9M development cost for stations and their integration into the DSN might not be required if previous missions had already implemented such a beacon mode. The additional cost for operating the beacon mode is estimated to be $600k/year. These costs are not included in the cost estimates provided in Section VII.

VI. **Solar Sail**

The technical feasibility study described in the following section also requires assumptions about the nature and performance of the solar sail. The sail itself consists of a thin (≤2 µm) plastic film, such as kapton, with highly a reflecting coating on the front (toward the Sun) and a coating of thermally emitting material on the back. The strawman design for the solar polar mission is a square sail, 150 to 200 m on a side. The spacecraft is located at a hole in the center of the sail which keeps it from being overheated by reflection from the sail. A sketch of a possible configuration is shown in Figure 6.1.

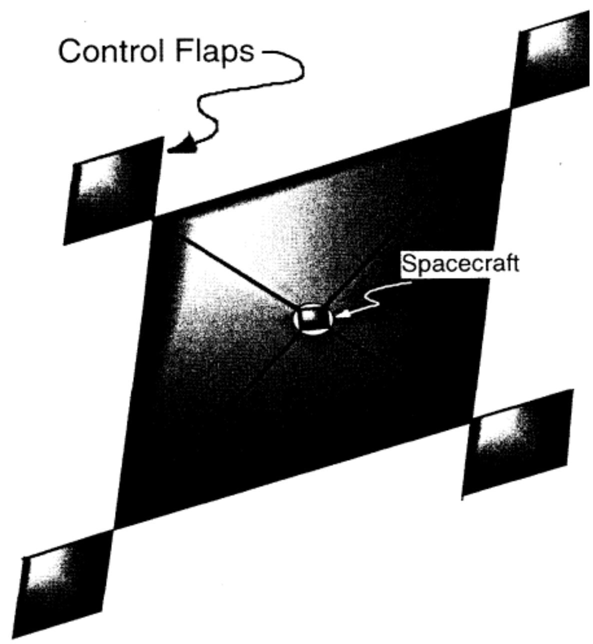

Figure 6.1. Sketch of a candidate configuration of a solar sail for SPSM.

There are several concepts of how to control the orientation of the sail, including:

- Control of the center-of-mass with respect to the center-of-pressure. This method, which is accomplished with a 3-axis-stabilized spacecraft bus connected to a 2-axis gimbal located on the solar sail (illustrated in Figure 7.1), was assumed for the mission feasibility study (Section VII).
- Articulation of sail segments such that the necessary imbalance of forces is provided by reefing or furling one or more quadrants of the sail (perhaps on a roller)
- Articulation of control flaps on the corners of the sail (illustrated in Figure 6.1). One analysis (Spilker, 1996) indicates that the sail can be turned through 90° in less than an hour by feathering the flaps on one side while leaving the flaps on the other side facing the Sun.
- Surface reflectivity changes; the corners of the sail could be coated with an electrochromic material that changes its reflectivity in response to the application of an electric potential.
- Passive stabilization (camber in sail)
- Classical methods such as thrusters

In addition to the sail itself, there must also be some support structure (spars) and mechanisms for deploying the sail at the start of the mission. Table 6.1 summarizes the mass breakdown of a design (generated by DLR, Germany) for a 150 m x 150 m sail with carbon fiber booms at 100 g/m.

TABLE 6.1. MASS BREAKDOWN OF ONE DESIGN OF A SOLAR SAIL.

Component	**Mass (kg)**
Film (2 µm)	79
Booms (4 @ 106 m)	43
Deployment system	15
Stowage canister	15
Total	152
Loading factor	6.8 g/m^2

The sail would be jettisoned once the final circular polar orbit is reached.

VI. **Technical Feasibility Study**

A. Approach

A technical feasibility study was carried out by JPL's Advanced Projects Design Team (unofficially known as Team X) on March 11-14 and July 11, 1997. Team X members represent each of the disciplines needed to establish the feasibility of a new mission concept and to estimate its cost; the roster for the SPSM study is given in Appendix C. At the first Team-X session, input information was provided about the science requirements, the trajectory design, and the solar sail parameters. The spacecraft was designed and its costs were estimated during the follow-on sessions. During each session Team X and the SPSM science team were together in the same room and each Team X member had electronic access to the relevant data bases and to each other.

B. Requirements Summary

The principal science requirements are given in Section IV, especially Section IV.F, while requirements arising from mission design and the use of the solar sail are given in Sections III and VI, respectively. The following system-level requirements were also agreed on:

- Launch date (determines technology available): 2005
- Mission duration: ≤7 years (cruise + on-orbit operations)
- Mission class: B/C
- Hardware models: Protoflight spacecraft and protoflight instruments
- Redundancy: Selected
- Spares: Selected
- Parts class: Class B, Mil-883B
- Spacecraft Supplier: JPL, based on X2000 technology

- Instrument Supplier: Various
- Integration and Test Site: JPL
- Launch Site: ETR
- Data Latency: ≤ 1 week
- Cruise Science: Not to be considered in designing the spacecraft or costing the mission
- Contingencies on mass and power: 20% on science; 30% on dry spacecraft

C. Flight System Architecture

Figure 7.1 shows a possible configuration for the SPSM spacecraft. Once in the final polar orbit, the solar sail and its booms, container, and control boom would be jettisoned, leaving only the rather simple structures seen in the lower part of the figures.

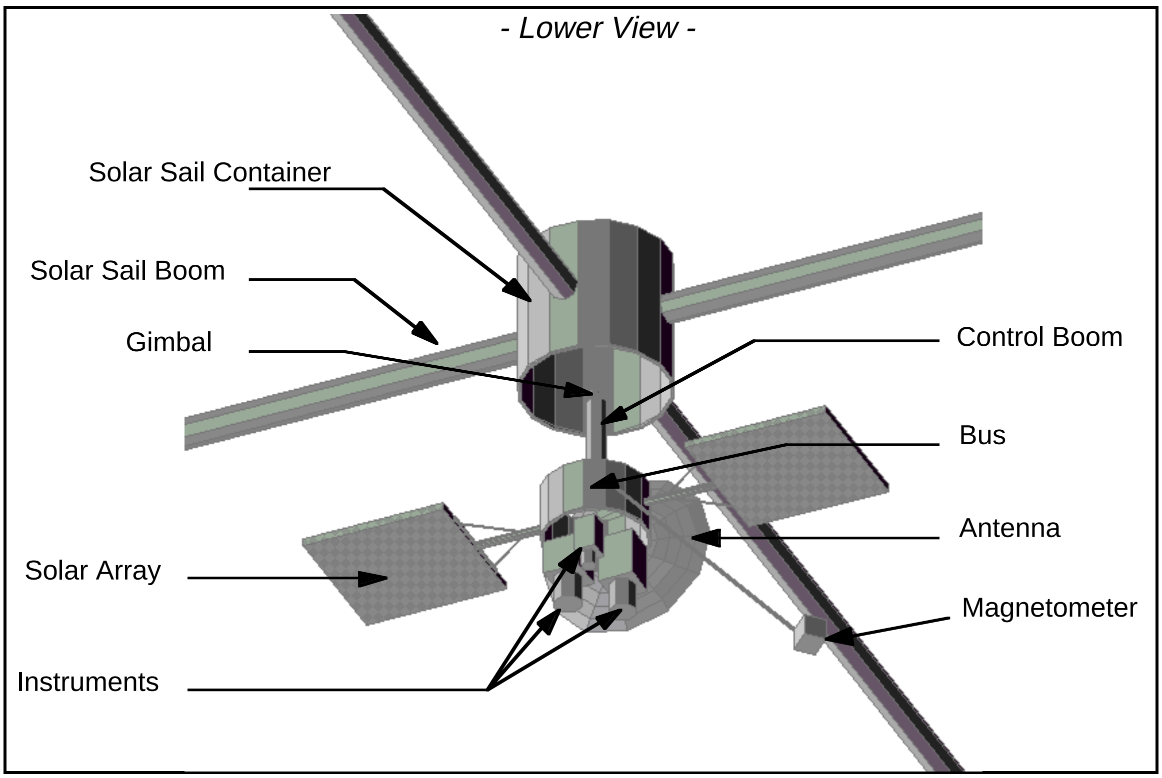

Figure 7.1. Sketch of a possible spacecraft configuration for SPSM.

D. Subsystems

Some of the spacecraft subsystem parameters that emerged from the Team X study can be summarized as follows:

Attitude Control Subsystem (ACS). The attitude control system must be capable of meeting the stringent requirements of SPSM's remote sensing instruments (Table 4.6). The approach selected incorporates sun sensors, star cameras, gyros, and reaction wheels.

The ACS provides coarse digital sunsensors which place the Sun in the coronagraph field of view, while the coronagraph is responsible for knowledge of fine pointing with respect to the Sun line. To meet the lifetime requirements for SPSM, the design has block redundant coarse sun sensors, star cameras, inertial reference units, interface electronics, propulsion valve drive electronics, and the sail control interface (which is not currently defined). There are internally or functionally redundant reaction wheels (4, with only 3 needed), wheel drive electronics, and single-axis drive actuator for the solar array.

Propulsion Subsystem: A hydrazine propulsion system is used to unload the reaction wheels and to maneuver the spacecraft away from the solar sail on reaching the final orbit. The wet mass of the propulsion subsystem is 22 kg.

Command and Data Handling Subsystem (C&DH). The C&DH is a block redundant system that collects data from the instruments, compresses it, stores it, and then prepares it for telemetry. The C&DH also controls critical spacecraft functions such as performing the attitude determination and control functions and decoding the uplink packets. It is block redundant.

Power Subsystem. The power for SPSM is provided by solar panels based on gallium arsenide solar array technology with a surface area of 1.1 m^2 There is also an advanced secondary Li-ion battery which will be used during launch, communications sessions, and coarse correction maneuvers.

Thermal Control Subsystem. The thermal control is basically passive, using electric heaters/thermostats to control sensitive spacecraft elements. The temperatures expected at 0.48 AU are within the qualification levels of most thermal control elements.

Structure Subsystem. A JPL in-house special-purpose design was assumed in order to save mass compared to a less expensive but probably heavier general-purpose spacecraft bus procured from industry.

Telecommunications Subsystem. The length of the SPSM mission calls for a fully redundant telecom system, except for the antennas. The principal link for data return is a body-fixed, 1.5 m antenna operating at X-band radiating 13 w RF power to the DSN 70 m antenna. The data rate is 91 kbps and the link has a 3 dB data margin at a range of 1 AU from Earth. The data accumulated at an average rate of ~4 kbps can be returned in a single 8-hour pass per week. The spacecraft also has an X-band low rate link to be used during launch, cruise, and emergencies; this link is provided by three omni antennas. It supports a bit rate of ~12 bps.

E. System Summary and Cost

Table 7.1 displays the system level mass and power summary that resulted from the Team X analysis. The total system mass at launch is 380 kg, which can be accommodated by a Taurus XL with Star 37FM launch vehicle. The mass of the instrument payload is 30 kg + 6 kg contingency. Increasing the instrument mass above that level would have a deleterious effect on the flight time to the final orbit. The instrument power is not a strongly constrained commodity because the instrument operation and most of the telecommunications occur at a solar distance of 0.48 AU.

TABLE 7.1. SYSTEM LEVEL SUMMARY OF MASS AND POWER.

SYSTEMS WORKSHEET — ***Solar Polar Sail Mission***

Analyst: J. A. Aguilar
Date: 3/11-12,14/97 ***Directory:*** TEAMX:ACTIVE STUDIES:OTHER STUDIES:SolarPolarSail:NEW_SolarPolarSail_XL

Legend: Inputs from Subsystems; Inputs from Systems; Calculated

Stabilization - cruise	***3-Axis***	Pointing Direction - cruise	***N/A***
Stabilization - science	***3-Axis***	Pointing Direction - science	***Solar***

P/L Pointing Control (arcsec 3 sigma)	***30.0***	Radiation Total Dose (krad)	72		
P/L Pointing Knowledge (aresec 3 sigma)	***5.0***	Redundancy	Selected	P/L Data Rate (kbps)	4
P/L Stability (arcsec/sec 3 sigma)	***2.0***	Mission Duration (years)	5 year flight time)	Downlink Data Rate (kb/s)	***125***
Pointing Requirements Driven By:	***Payload***	Science BER	***1.00E-05***	Data Storage (Gb)	***10***

41.7	Mass (kg)	Mode 1 Power (W) Science	Mode 2 Power (W) Xmit	Mode 3 Power (W) Cruise	Mode 4 Power (W) Launch	Mode 5 Power (W) N/A	NASA TRL	Last Updated
Payload								
Instruments	30.0	33.5	3.2	3.2	0.0	0.0	5.3	3/17/97 10:05
Solar Sail *DLR (Everything Included)*	150.0	0.0	0.0	0.0	0.0	0.0		
Payload Total	180.0	33.5	3.2	3.2	0.0	0.0		
Bus								
Attitude Control	15.9	29.8	29.8	29.8	16.9	0.0	4.0	3/14/97 16:18
Command & Data	10.4	25.4	25.4	25.4	25.4	0.0	3.0	3/14/97 16:03
Power	10.4	15.8	16.1	14.0	16.7	2.2	4.0	3/14/97 15:54
Propulsion	8.7	4.0	4.0	4.0	41.0	0.0	7.0	3/14/97 14:48
Structure	54.1	0.0	0.0	0.0	0.0	0.0	6.0	3/14/97 16:15
S/C Adapter	4.2							
Cabling	13.3							3/14/97 16:15
Telecomm	13.1	12.0	44.5	28.3	28.3	0.0	5.0	3/14/97 16:11
Thermal	4.4	17.0	17.0	17.0	17.0	17.0	7.0	3/14/97 14:29
Bus Total	134.4	104.0	136.8	118.4	145.2	19.2		
Spacecraft Total (Dry)	314.4	137.5	140.0	121.6	145.2	19.2		
Mass/Power Contingency	46.3	37.9	41.7	36.2	43.6	5.8		
Spacecraft with Contingency	***360.7***	***175.4***	***181.7***	***157.8***	***188.7***	***25.0***		
Propellant & Pressurant	4.8	For S/C mass =	***240.0***					3/14/97 14:48
Spacecraft Total (Wet)	***365.5***							
L/V Adapter	14.6	*Includes 10% contingency*						3/14/97 16:15
Launch Mass	***380.1***	230.1 (mass w/o solar sail)						
Launch Vehicle Capability	***421.0***	***Taurus XL w/ Star 37FM***		Launch C3	0.5		9.0	3/12/97 9:10
				Fairing type	standard			
				Fairing dia., m	1.37			
Launch Vehicle Margin	***40.9***	***9.7%***						

Contingencies	Mass	Power
Instruments	20%	20%
Solar Sail	0%	0%
S/C, dry	30%	30%

Team X used its Deep Space Cost Model to estimate costs for this project. That model includes quasi-grass roots cost estimates for the spacecraft subsystems, the payload, mission operations, and the launch vehicle. Historical cost models are used for other mission components, including systems engineering, assembly, test, and launch operations (ATLO), project management, phases A and B, and reserves. The cost of the sail is based on an estimate from a potential vendor. Costs for developing advanced technology items (see Section VIII) are not included, nor is the cost of DSN tracking time. Although the costs are computed in uninflated FY97 dollars, the technology base and the durations of the several mission phases are based on the schedule given in Table 7.2.

TABLE 7.2. SCHEDULE ASSUMED FOR COSTING

Task	Duration	Start	Finish
Start		1/22/03	
Phase A	13.5 w	1/22/03	4/25/03
Phase B	26 w	4/28/03	10/24/03
Phase C/D	104 w	10/27/03	10/21/05
Launch		9/24/05	
Flight Operations	322 w	10/24/05	12/23/11

Table 7.3 provides the estimated total project cost in FY $97M.

As pointed out in Section III, Team X evaluated a mission with a 158 m square sail and a cruise time of 4.6 years. The cruise could be shortened to <4 years by using a larger sail (see Table 3.2). While the cost of operations would be reduced, the launch vehicle cost would increase from $38M to $47M and the cost of the sail might increase from ~$10M to perhaps $20M.

TABLE 7.3 ESTIMATE OF COST OF SPSM MISSION.

Phases A/B/C/D		
1.0 Project Management & Outreach		$4.6M
2.0 Science		2.0
3.0 Project & Mission Engineering		2.3
4.0 Payload		46.7
4.1 Management	(0.5)	
4.2 Payload Engineering	(1.0)	
4.3 Instruments	(42.0)	
4.4 Integration & test support	(3.4)	
5.0 Spacecraft		70.9
5.1 S/C System Management	(0.6)	
5.2 S/C System Engineering	(1.0)	
5.3 Subsystems		
Attitude Control	(17.8)	
` Command and Data	(8.5)	
Telecommunications	(10.2)	
Power	(4.6)	
Solar Sail	(9.6)	
Other propulsion	(4.1)	
Structure, mechanisms, cabling	(8.9)	
S/C Mechanical Buildup	(2.8)	
Thermal Control	(1.7)	
Software	(1.0)	
6.0 Assembly, Test, and Launch Operation		4.8
7.0 Mission Operations		10.0

Subtotal		141.2
Reserves @ 20%		28.2
Launch Vehicle (Taurus XL/Star 37)		38.0

Total Phases A/B/C/D		$207.4M
Phase E		
1.0 Project Management and Outreach		3.0
2.0 Science		26.0
3.0 Mission Operations		23.0

Subtotal		52.0
Reserves @ 10%		5.2

Total Phase E		$57.2M
Probable Total Cost of Project	**$265M**	
+20% (exclusive of LV & reserves)	303	
- 20% (exclusive of LV & reserves)	226	

VIII. **Technology Development**

The Solar Polar Sail Mission is critically dependent on the successful development of solar sails. Technology that requires development and demonstration includes:

- Construction of affordable sails in the 150-200 m square size range
- Achievement of loading factors in the neighborhood of 6 g/m^2, preferably less.
- Successful deployment in space of a sail in the 150-200 m square size range.
- Control of the sail by the sail itself, whether it be by center of pressure versus center of mass, control flaps, electrochromic variations, furling, or other means.
- Long-term maintenance of reflectivity and thermal properties. Since the SPSM has no planetary encounters or other critical events, however, the mission can still be successful if there is some degradation in those properties; it will just take longer to reach the final orbit.
- Software for navigation (including Earth and planetary perturbations) and sail control.

Two flight validation tests are in the planning stages. The first would test a small (30-50 m square) sail with a relatively large sail loading factor (20 g/m^2) on which the control of the sailcraft attitude would be performed using the spacecraft's cold-gas attitude control system but with the spacecraft separated from the sail by a boom. Sailcraft attitude control using the center-of-mass versus center-of-pressure technique would be carried out as an experiment. The second flight validation test would be closer to what is needed for SPSM: sail loading = 10 g/m^2, ~100 m on a side, a lightweight mechanical deployment system, and some type of photon-pressure sail control.

Substantial software development is also required for a solar sail mission, including sail control modeling and algorithms, low-thrust trajectory simulations, and navigation.

Aside from the solar sail, a few subsystem items were included in the feasibility study which are not currently funded as part of the X2000 or other programs for readiness by the start of 2003. The list includes: miniaturized reaction wheels (modified commercial reaction wheels), multi-chip module gimbal drive electronics, micro-machined silicon vibratory gyroscopes (currrently removed from the X2000 baseline), and the tiny deep space transponder (current technology cutoff date of 2003).

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Appendix B. Trajectory Plots

A general idea of some of the trajectory parameters can be obtained by considering a fixed value of the "sail characteristic acceleration" (SCA) which is the acceleration at a distance from the Sun of 1 AU. The time profiles of many trajectory parameters, such as the flight time, distance from Sun, and distance from the ecliptic approximately scale with SCA; however, other parameters, such as angles and distances with respect to Earth, must be calculated separately for each value of SCA. The plots shown here are based on the assumption that SCA = 1 mm/s^2, which is higher than the more realistic 0.716 mm/s^2 value appropriate for a 200 m square sail with a sail loading of 6.8 g/m^2.

Figure B.1 shows an approximately steady increase in ecliptic latitude from launch through arrival 1050 days (2.88 years) later. The time scale should be multiplied by 1/0.716 = 1.4 for SCA = 0.716. The time can also be decreased by 83/90 if the launch date is restricted to be near the equinoxes when the Earth is at its maximum distance above or below the solar equator.

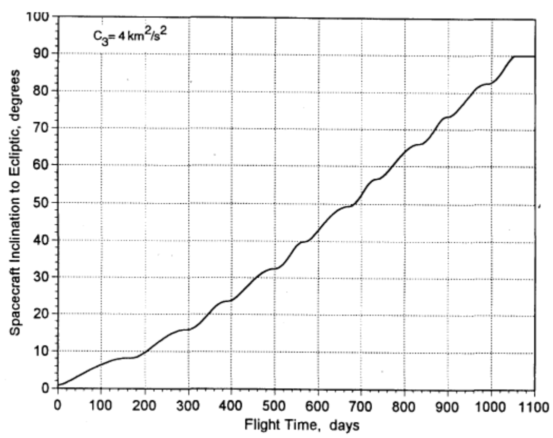

Figure B.1. Spacecraft inclination to the ecliptic versus time assuming SCA = 1 mm/s^2.

Figure B.2 shows the thrust vector cone angle. The plane of the sail is at right angles to the thrust vector and the angle between the sail plane and the solar direction is ~56° for much of the flight.

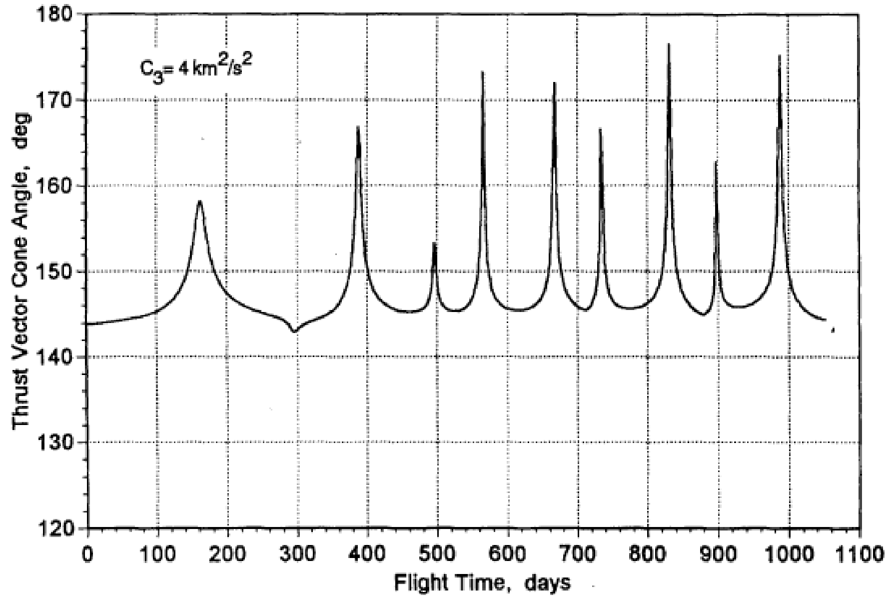

Figure B.2. Sun-spacecraft distance versus time for SCA = 1 mm/s^2.

Figure B.3 indicates the Sun-spacecraft distance during cruise to the final orbit, after which the distance remains fixed at 0.48 AU.

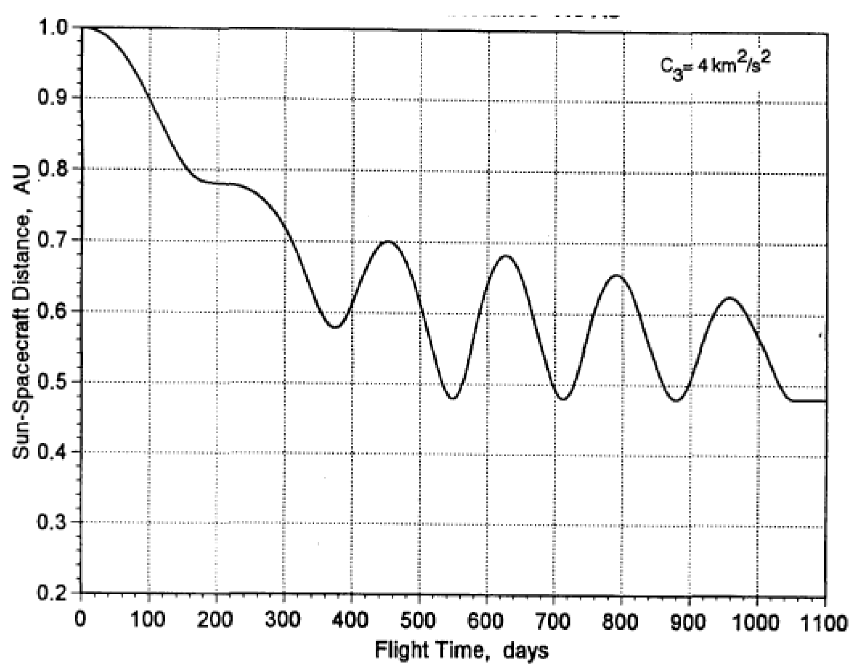

Figure B.3 Sun-spacecraft distance versus time for SCA = 1.0 mm/s^2

Figures B.4 and B.5 show the Earth-spacecraft separation and the Earth-Sun-spacecraft angle for both cruise and the final orbits. The distance of the spacecraft from Earth never exceeds 1.7 AU during cruise and remains <1.45 AU after the final orbit is reached; the communications system must be designed to cover this range. From Figure B.5 it can be seen that there is only a very short interval during cruise that the ray path from Earth to the spacecraft comes within 10° of the Sun, so solar conjunctions are not a problem for communications.

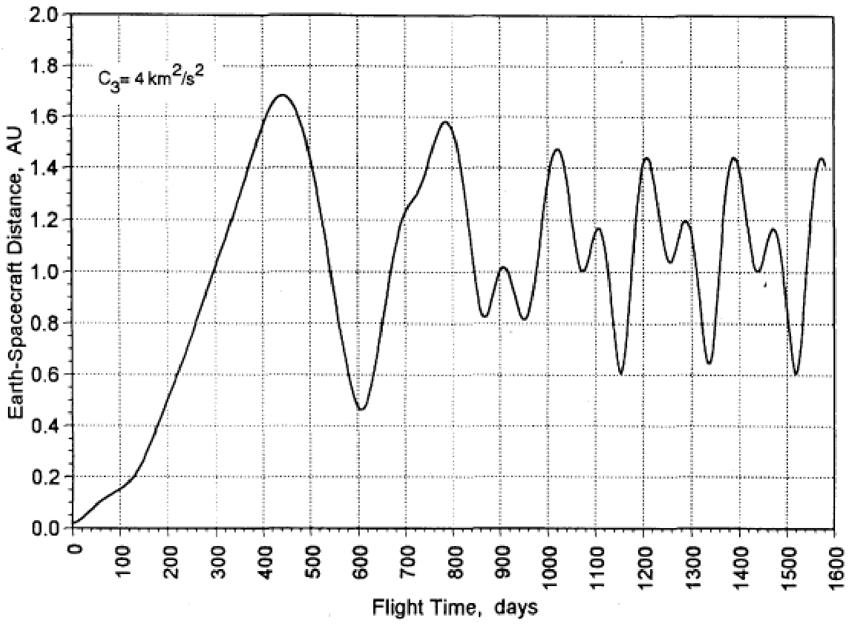

Figure B.4. Earth-spacecraft distance versus flight time for a trajectory with SCA = 1 mm/s^2.

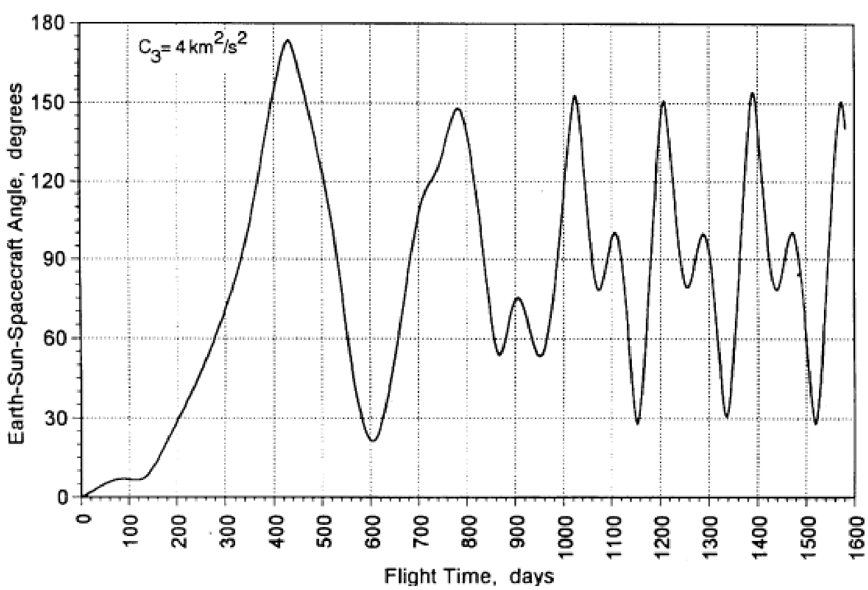

Figure B.5 Thrust vector cone angle versus time for a trajectory with SCA = 1 mm/s^2.

Appendix C. Team X Members for SPSM Study

Subsystem	**Team Members**
Study Leader	Richard Bennett
Science	Tom Spilker
Mission Design	George Carlisle
Ground Systems	Mark Rokey
Systems	George Sprague
Instruments	Jim Anderson
Propulsion	Ron Klemetson
ACS	Ed Swenka/Ed Mettler
CDS	Vince Randolph
Power	Steve Dawson
Thermal	Bob Miyake
Structures	Gerhard Klose
Telecom-System	Anil Kantak
Telecom-Hardware	Faiza Lansing
Programmatics	Ralph Bartera
Cost	Keith Warfield
Documentation	Larry Palkovic